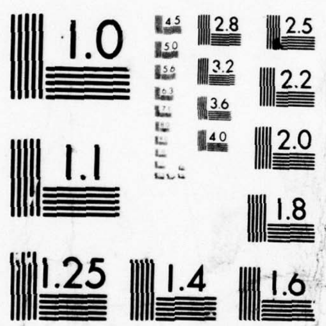
AD-A066 477 FOREIGN TECHNOLOGY DIV WRIGHT-PATTERSON AFB OHIO F/G 20/4
EXPERIMENTAL STUDY OF THE SUPERSONIC FLOW AROUND V-SHAPED WINGS--ETC(U)
AUG 78 A L GONOR, M N KAZAKOV, A I SHVETS

UNCLASSIFIED FTD-ID(RS)T-1017-78 NL

| OF |

AD
A066477

END
DATE
FILMED
5--79
DDC



MICROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

AD-A066 477

FTD-ID(RS)T-1017-78

(1)

FOREIGN TECHNOLOGY DIVISION

EXPERIMENTAL STUDY OF THE SUPERSONIC FLOW AROUND V-SHAPED WINGS

By

A. L. Gonor, M. N. Kazakov, A. I. Shvets

FTD- ID(RS)T-1017-78

UNEDITED MACHINE TRANSLATION

FTD-ID(RS)T-1017-78 9 August 1978

MICROFICHE NR: FTD-78-C-001089L

EXPERIMENTAL STUDY OF THE SUPERSONIC FLOW AROUND V-SHAPED WINGS

By: A. L. Gonor, M. N. Kazakov, A. I. Shvets

English pages: 26

Source: Trudy Nauchnoy Mekhanicheskiy Institut Universiteta Moskva, Nr. 1, 1970, pp. 58-70

Country of Origin: USSR
This document is a machine translation.

Requester: FTD/TQTA
Approved for public release; distribution unlimited.

THIS TRANSLATION IS A RENDITION OF THE ORIGINAL FOREIGN TEXT WITHOUT ANY ANALYTICAL OR EDITORIAL COMMENT. STATEMENTS OR THEORIES ADVOCATED OR IMPLIED ARE THOSE OF THE SOURCE AND DO NOT NECESSARILY REFLECT THE POSITION OR OPINION OF THE FOREIGN TECHNOLOGY DIVISION.

PREPARED BY:

TRANSLATION DIVISION
FOREIGN TECHNOLOGY DIVISION
WP-AFB, OHIO.

U. S. BOARD ON GEOGRAPHIC NAMES TRANSLITERATION SYSTEM

Block	Italic	Transliteration	Block	Italic	Transliteration
А а	*А а*	A, a	Р р	*Р р*	R, r
Б б	*Б б*	B, b	С с	*С с*	S, s
В в	*В в*	V, v	Т т	*Т т*	T, t
Г г	*Г г*	G, g	У у	*У у*	U, u
Д д	*Д д*	D, d	Ф ф	*Ф ф*	F, f
Е е	*Е е*	Ye, ye; E, e*	Х х	*Х х*	Kh, kh
Ж ж	*Ж ж*	Zh, zh	Ц ц	*Ц ц*	Ts, ts
З з	*З з*	Z, z	Ч ч	*Ч ч*	Ch, ch
И и	*И и*	I, i	Ш ш	*Ш ш*	Sh, sh
Й й	*Й й*	Y, y	Щ щ	*Щ щ*	Shch, shch
К к	*К к*	K, k	Ъ ъ	*Ъ ъ*	"
Л л	*Л л*	L, l	Ы ы	*Ы ы*	Y, y
М м	*М м*	M, m	Ь ь	*Ь ь*	'
Н н	*Н н*	N, n	Э э	*Э э*	E, e
О о	*О о*	O, o	Ю ю	*Ю ю*	Yu, yu
П п	*П п*	P, p	Я я	*Я я*	Ya, ya

*ye initially, after vowels, and after ъ, ь; e elsewhere.
When written as ё in Russian, transliterate as yё or ё.

RUSSIAN AND ENGLISH TRIGONOMETRIC FUNCTIONS

Russian	English	Russian	English	Russian	English
sin	sin	sh	sinh	arc sh	$\sinh^{-1}$
cos	cos	ch	cosh	arc ch	$\cosh^{-1}$
tg	tan	th	tanh	arc th	$\tanh^{-1}$
ctg	cot	cth	coth	arc cth	$\coth^{-1}$
sec	sec	sch	sech	arc sch	sech^{-1}
cosec	csc	csch	csch	arc csch	csch^{-1}

Russian	English
rot	curl
lg	log

Page 58.

EXPERIMENTAL STUDY OF THE SUPERSONIC FLOW AROUND V-SHAPED WINGS.

A. L. Gonor, M. N. Kazakov, A. I. Shvets.

A question concerning airfoil lift at the hypersonic speeds of flight attracts increasing attention of the researchers. In the literature are discussed the projects of the orbital flight vehicles, capable of completing descent from the trajectory of satellite, the manuvering flight in the atmosphere and touchdown with the aid of wings with variable geometry [1]. Actual/urgent becomes the problem of the creation of hypersonic aircraft.

In connection with the tasks indicated in recent years, will arise considerable interest in the study of flow around of the wings at high supersonic velocities. Investigations are conducted in two main directions: is studied the hypersonic flow around of the wings of traditional form (fine/thin wings of triangular and of other planforms) is conducted the search of the new forms of the layouts of these possessing optimum aerodynamic or thermal characteristics. The

last/latter direction includes numerous studies of flow around of the wings with V-shaped cross section. Initially was studied the special class of V-shaped wings, corresponding to flow with the plane shock wave, connected to leading edges [2, 3]. To the experimental check of this simple diagram of flow and its modifications was dedicated the whole series of works [4-14].

Subsequently were found an additional two classes of the exact solutions, corresponding to flow around V-shaped wings with regular and Mach interaction of shock waves [15, 16]. Both the type of flows were observed experimentally [17 - 19].

By the important result of investigations V-shaped wings was the establishment of the high value of their lift-drag ratio [15], noticeably exceeding the maximum quality of equivalent in volume wing with elliptical cross section. The high value of lift-drag ratio was reveal/detected, also, in calculations of V-shaped wings, streamlined with plane wave [20]. One of the authors of present article was carried out calculation on the basis of Newton's theory taking into account concentrated forces [21], that showed that wing of V-shaped form during some limitations (for example, with the assigned lift and volume) has the maximum value of lift-drag ratio at expansion angles less than 180°. As illustration to this calculation Fig. 1 depicts the dependence of lift-drag ratio on the expansion angle of wing γ

under assigned lift C_y referred to wing area in plan/layout S, for several values of relative volume $\tau = V/S^{3/2}$. It is evident that the maximum value of quality corresponds to the expansion angle of ~140°.

This work is dedicated to the experimental study of flow around V-shaped wings over a wide range of a change in the geometric parameters. On the basis of the obtained results, are analyzed the possible patterns of flow around of the wings, is determined the distribution of pressure on the wing surface and the shock configuration. With the aid of weight measurements is located the dependence of lift-drag ratio on expansion angle of V-shaped wing.

Page 59.

In the first group of experiments, was studied the flow about the drain/vented models of V-shaped wings during a change in the expansion angle of wings from 0° to 180°, angle of attack from 0° to 15° and number M=4. The first results of these experiments were published in work [19].

Construction made it possible to rotate one half of wing relative to another and to change with this expansion angle (Fig. 2). The removable parts of the wing, carried out in the form of triangular plates with pointed edges, could be removed and were repladed by others.

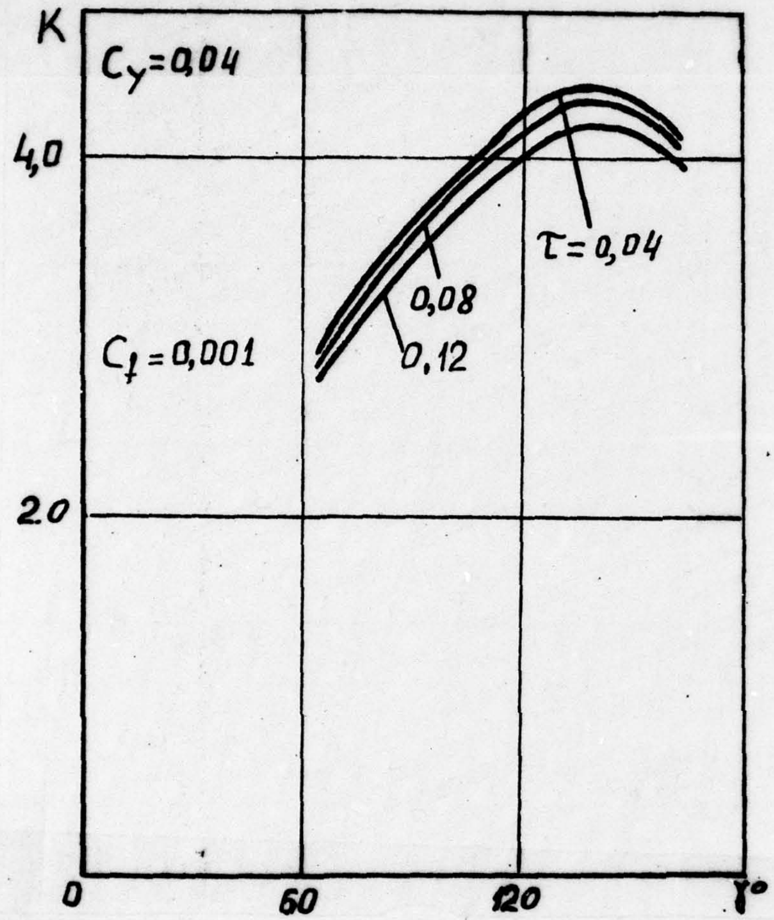


Fig. 1.

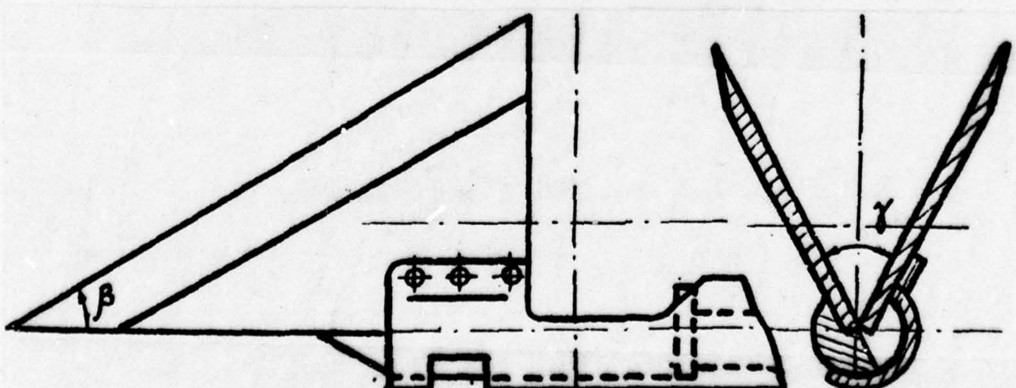


Fig. 2.

Page 60.

For testings were made three steel models of V-shaped wings with apex angles β, by the respectively equal to 30° - model No 1, 35° - No 2 and 40° - No 3. The length of trailing edge of all models of wings was identical and it was 335 mm.

For the study of flow pattern between wings after model, was establish/installed the comb/rack, measuring the total pressures. Tubes in comb/rack were arrange/located in the plane of the symmetry of wing at a distance of 20 mm from its rear section/shear. Model was establish/installed in the forward section of the optical window, and field of flow was studied on trace after model (Fig. 3). The position of jump system was determined by ratio h/R, where h - a distance from the axis of model to gallop in the plane of the symmetry of wings, measured in the section of rear section/shear, R - a distance in this same section from axis to wing edge. In Fig. 4 are constructed the graph/diagrams of the dependence of value h/R from expansion angle γ for model No 1. Each curve corresponds to the specific angle of attack. Unbroken curve are connected the experimental points, corresponding in the photographs of the brightest intersection of jumps, In series of experiments, were observed several intersections

of the jumps of those noted in Fig. 4 by small circles. Dot-dash curved is shown the position of the projection of leading wing edge in the plane of symmetry during a change in the expansion angle. With the approach of wings, shock wave gradually will move away from axis.

It is of interest to explain, which flow conditions are realized during the consecutive decrease of expansion angle, beginning from flat/plane delta wing (γ=180°) and terminating understand by the complete approach of wings (γ=0). The interpretation of photographs showed that are realized several types of flow with different shock envelopes (Fig. 5).

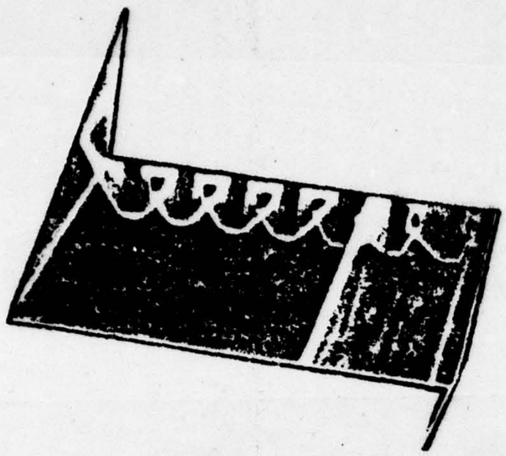


Fig. 3.

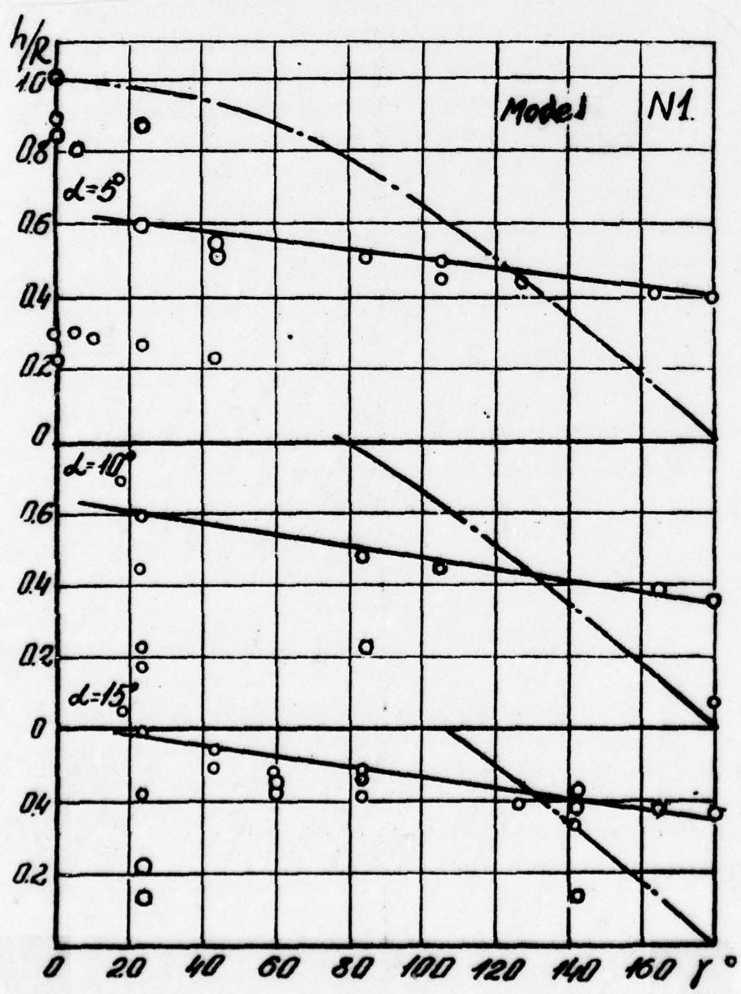

h/R
Model N1.
α=5°
α=10°
α=15°
1.0
0.8
0.6
0.4
0.2
0
0
20
40
60
80
100
120
140
160
γ°

Fig. 4.

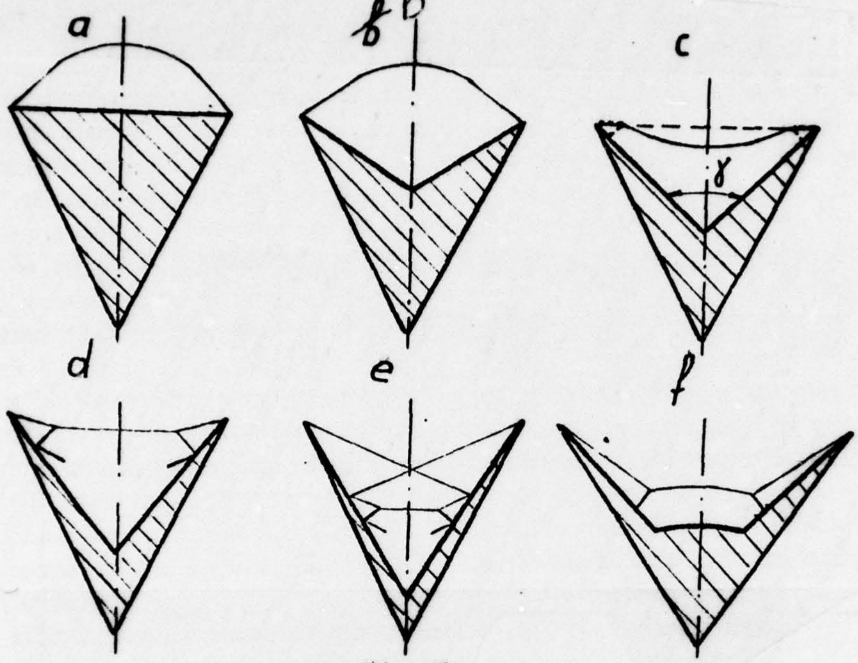


Fig. 5.

At large expansion angles (beginning from 180°) flow it occurs with one leading wave, connected to leading wing edges (a). With it decreases, and it is moved nearer to the surface of wing (b). Further decrease of angle γ leads to the pattern of flow with almost plane wave (c), moreover at the special values of the parameters wave stops in the accuracy of plane. Let us note, that the interpretation of the photographs, which correspond to pattern (c), allow/assumes two possible systems of jumps. One of them is shown by solid line, another - a dotted line. Flow conditions when jump system wholly is between wings, are shown on Fig. 5d and 5e.

The type of flow, following it is direct after flow with almost step shock (Fig. 5c), it has the three-dimensional/space configuration of Mach type jumps. For some wings is demonstrated the possibility of flow with the Mach system of step shocks [22]. In the general case the jumps are bent and, besides Mach system, are present the jumps reflected, which interact between themselves.

Figure 5e, corresponds to the regular intersection of jumps. This picture of flow displaces the Mach configuration of waves during further approach of wings. Here just as in the preceding case, appears the complex system of jumps reflected and secondary interactions. According to the shadowgraphs, the flow of the type indicated is retained to the very low values of angle γ (order of 3°), after which under conditions of the carried out experiments the range between wings is filled with boundary layer, and supersonic flow is destroyed. During, transition to other models of the pattern of flows, besides angle γ, they will depend substantially from one more parameter as which it is possible to take relation to the cross-sectional area of model (shaded range in Fig. 5) to the area of the clearance between wings in the plane of normal velocity of incident flow. Let us designate this parameter through λ. At preset angle γ, the value of the parameter λ for all models changed from 0 ($\alpha=0^\circ$) to certain value, corresponding to the angle of attack of $\alpha=15^\circ$.

According to experiments, at angles of $130° \leq \gamma < 180°$ topological structure of flow was retained for all λ from this range. At smaller angles γ with an increase in the parameter λ, additionally appeared the jumps repeatedly reflected from walls. Experiments showed that in the range of low values λ with increase the last/latter system of the jumps, limiting the range of disturbed flow, approaches a body surface. However, beginning with certain value λ [18], is realized inverse dependence. With the assigned Mach number, one or the other pattern of flow depending on values λ will be realized in the substantially different range of angles γ. In particular, the comparison of data shows that the decrease of the values of the parameter λ makes it possible to increase the range of expansion angles γ, as a result of which occurs of flow of the type 5d and 5e. Everything said is related to the case of the leading wave, connected to leading edge.

In the process of experiment, together with optical investigation, were conducted the measurements of the distribution of pressure on the surface of models and distributions of the total pressure in trace. If we designate distance from the axis of model to drainage point in this section through z and distance in this section from axis to wing edge through R, then constant value z/R

will correspond to points, by the lying/horizontal on one straight line, passing through the apex/vertex of models.

Page 63.

Figure 6 gives the experimental values of the coefficient of pressure C_p for model No 2 depending on value τ. On each curve/graph are constructed the curves C_p for the angles of attack of α=5°, 10°, 15° at the fixed value of expansion angle γ. Let us observe the behavior of these curved during the successive approximation wings from γ=180° to 20°.

During, the complete deployment of wings and low angles of attack, the pressure on spread/scope is retained approximately constant. An increase in the angle of attack leads to pressure rise on wing, moreover at angles α=10 and 15° curve C_p consists of two sections: the section of the elevated pressure, which begins from leading edge, and the section of reduced pressure about the plane of the symmetry of wings. Similar distribution of pressure is observed to angle into expansion/disclosures γ=160° (Fig. 6).

Further approach of wings qualitatively changes the character of pressure distribution. In the range of angles of γ=20-120° (Fig. 6) the distribution curve of pressure consists of several steps of

constant pressure, arrange/located on section the adjacent the edge, transition section and section near the plane of the symmetry of wing.

On the whole, the effect of expansion angle on pressure distribution is convenient to trace in Fig. 7, where are represented the distribution curves of pressure for model No. 2 at the angle of attack of α=15°.

The analysis of experimental data also showed that in the zone of minimum pressure, adjacent to edge, the expansion/disclosure of wings evenly raises pressure, and if at the angle of attack of α=5° this increase oscillates within limits ($0,005< C_p <0,07$), then at the angle of attack of α=15° value of the coefficient of pressure grew/rises from 0.02 to 0.25.

Page 64.

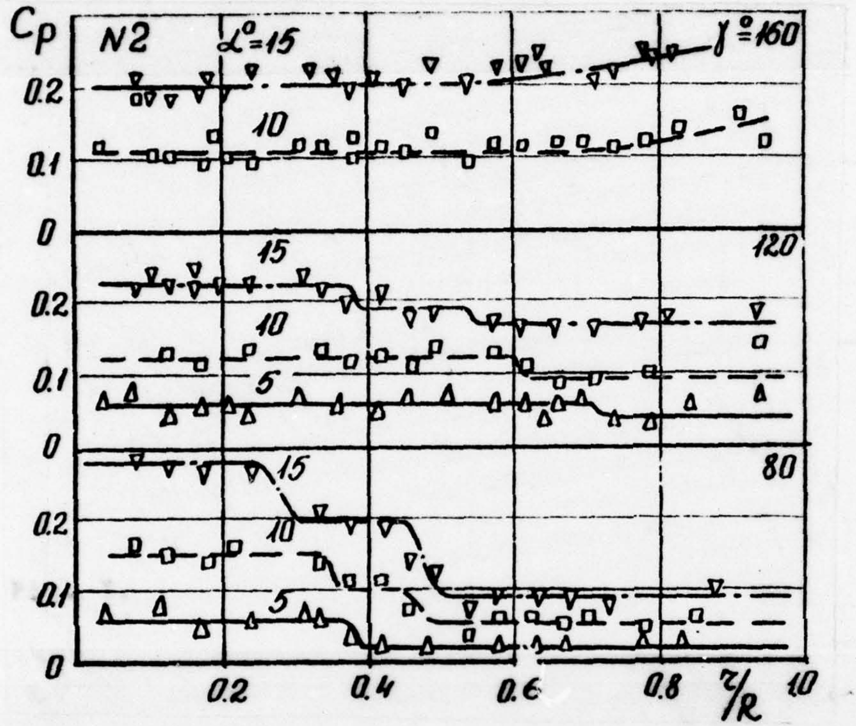


Fig. 6.

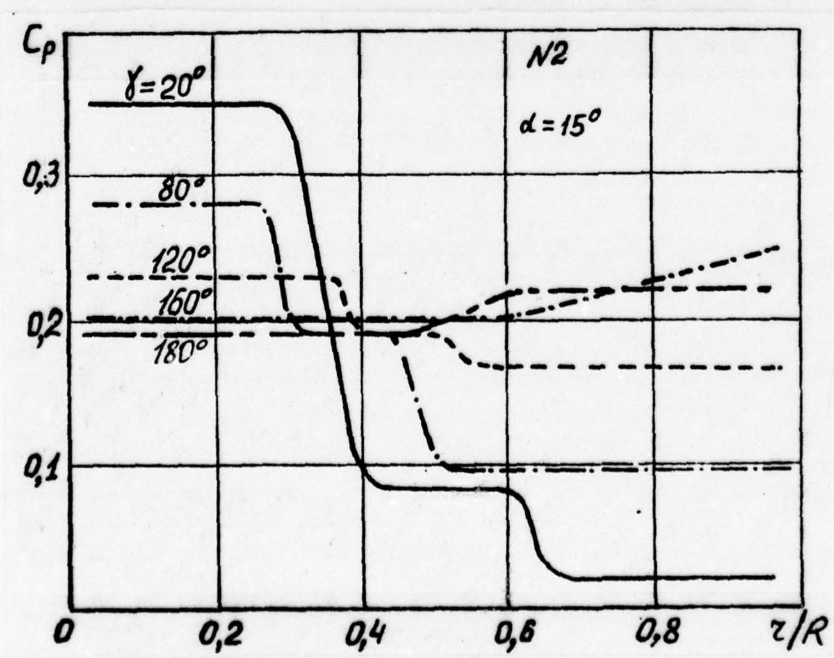


Fig. 7.

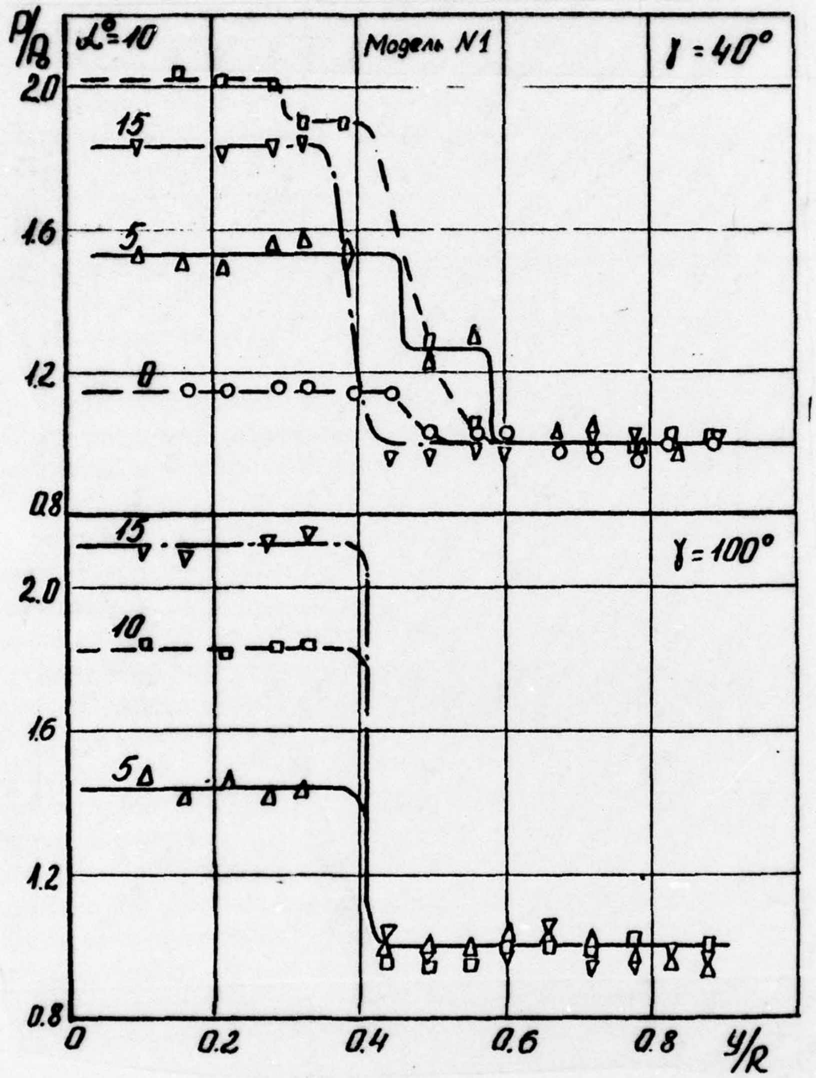


Fig. 8.

Page 65.

With the approach of wings, the extent of high-pressure zone

decreases, and the value of maximum pressure monotonically increases. The most essential pressure increment occurs at the angle of attack of α=15°, pressure coefficient in this case reaches values C_p =0,35.

Let us now move on to the short characteristic of field of flow in the plane of the symmetry of wings which was measured with the aid of the comb/rack of the total pressure. Taking into account that directly for model it is retained conical flow, can be on the measurements total pressures in trace judged their values in the range between wings.

The values of the pressures, measured by comb/rack, they were expressed in dimensionless form in the form P/p_0 where P - the total pressure, measured by tube; P/p_0 - the total pressure after normal shock in the undisturbed flow. Figures 8 gives the dependences of relation P/p_0 on value y/R (model 1). Here y - distance from the axis of model to tube in the section of comb/rack; R - distance in this same section from axis to the continuation of wing edge.

The location of experimental points indicates that at the large expansion angles of models (to γ=80°) the pressure in the plane of the symmetry of wings is changed abruptly, taking different constant values on individual sections. The first low-pressure section corresponds to the range of the undisturbed flow, the second section

cf elevated pressure is fcrmed as a result cf ccoperating the shock waves and reaches the axis cf mcdel. With the rest of angle of attack, the pressure on the second section is raised.

At small expansion angles, pressure distribution qualitatively retains its form; however, high-pressure area decomposes on several sections of constant pressure.

Compare the now findings cn pressure distribution and on the field of the total pressures with the optical picture of flow, described above. Let us examine first the large expansion angles of wings. Then, according tc shadow photographs, is only one leading shock wave, connected tc wing edges. Within flow there are no visible jumps. This conclusion/derivation is confirmed by smooth curve of the distribution of pressure cn wall, and also by diagram/curve of the total pressure in the plane of the symmetry of flow.

At expansion angles less than 120-140°, is realized the Mach configuration of jumps in space. It is real/actual, the field of the total pressures in the plane of the symmetry of wings corresponds to the flow, passed through one jump, and the distribution of pressure cn wall represents curve, that consists of two sections of constant pressure with the sharp transition, which corresponds to the leg of wave, arrange/located within flow. The seccndary jumps, reflected

frcm wall, weak and do not give a noticeable change in the pressure and entropy.

At expansion angles γ less than 80°, essential become secondary reflections from the walls which are exhibited noticeably on the curve/graph of the distribution of pressure. The number of sections of constant pressure now becomes more than two and grow/rises during decrease γ. The future of the expansion angle of wings leads to the replacement of Mach interaction of jumps regular. Conclusion this escape/ensues from the analysis of the disturbed field of the total pressures, in which the pressure recovery proves to be above than the theoretical optimum pattern, which contains by the scythe also of the normal shock.

Page 66.

It is interesting to note that for small angles γ (γ=20°) in range it is lower than the intersection of the jumps, which go from leading edges, the noticeable increases of the total pressure reveal/detected and only secondary and subsequent intersections of jumps form the zones of the high values of the quantity indicated. This special feature/peculiarity appears in connection with the decrease of the intensity of the connected to leading edges jumps during the decrease of angle γ.

In experiments the field of the total pressures was determined for purpose of the refinement of the picture of the flow between wings. However, these data have independent interest in connection with the possible use of compression of flow on lifting surfaces as the preliminary stage of air diffusion before the entrance into jet engine. The results of experiment showed that on tested V-shaped wings recovery factor higher than in the pattern of optimum plane diffuser with oblique and normal shocks although, apparently, is still distant from the optimum value which it is possible to achieve by the special selection of the geometric parameters of wings.

Let us now move on to the examination of experiments in the measurement of the forces, which act on V-shaped wings, with the aid of aerodynamic balances. Models for this series of experimental experiments (Fig. 9) represented the combination of fine/thin central cone (with the half-angle equal to 7°) with two rigidly attached on it delta wings (angle between the leading wing edge and the axis of cone was 30°). Testings were conducted on three-component strain-gauge balances by standard procedure. The aerodynamic characteristics of models are obtained in the range of the angles of attack of α=1-15°. The accuracy of experiments (rms error) during the use of the three-component of strain scales indicated comprised:

$$|\sigma_{c_n}|_{max} = 3\div 5\%,\ |\sigma_{c_\tau}| = 4\div 6\%,\ |\sigma_{c_g}| = 0{,}7\div 0{,}8\%.$$

Here C_n - the coefficient of normal force, C_τ - the coefficient of force of periphery, C_d - the center-of-pressure coefficient. Experiments were conducted at the velocity of incident flow, corresponding to number M=6.

Tested 4 models with the expansion angles of V-shaped wing of γ=180°, 170°, 160°, 150°. It should be noted that the pattern of flow about these models will not correspond, described it is above, for large expansion angles on two reasons: due to the bracket, prepared in the form of slender cone and hypersonic velocity of incident flow. As a result the type of flow will correspond that shown on Fig. 5f.

In the second group of experiments, were measured normal and axial components of the total aerodynamic force, acting on the model, and the pitching moment in the coordinate system, connected with weights. According to the results of measurements, were designed the coefficients of the frontal C_x and lift C_y of forces, the lift-drag ratio K and the center-of-pressure location models.

Test data are represented in Fig. 10 in the form of dependences $C_x, C_y = f(\alpha)$. During the calculation of aerodynamic coefficients C_x and C_y for characteristic area, was accepted the area of the

midsection of model. It should be noted that the model with angle of $\gamma = 170°$ has larger resistance than delta wing. This is connected with the fact that during the production of this model were allowed the deviations from intended sizes.

Page 67.

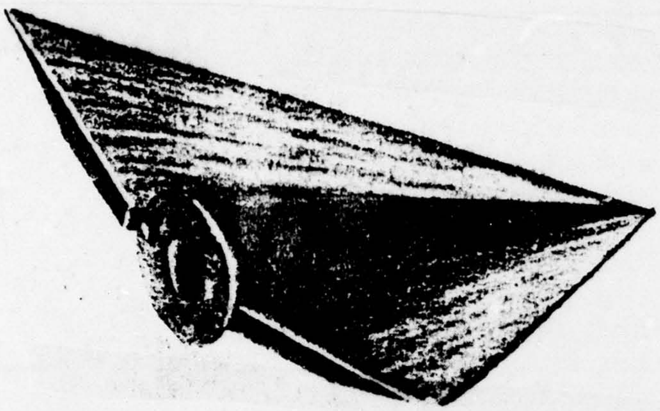


Fig. 9.

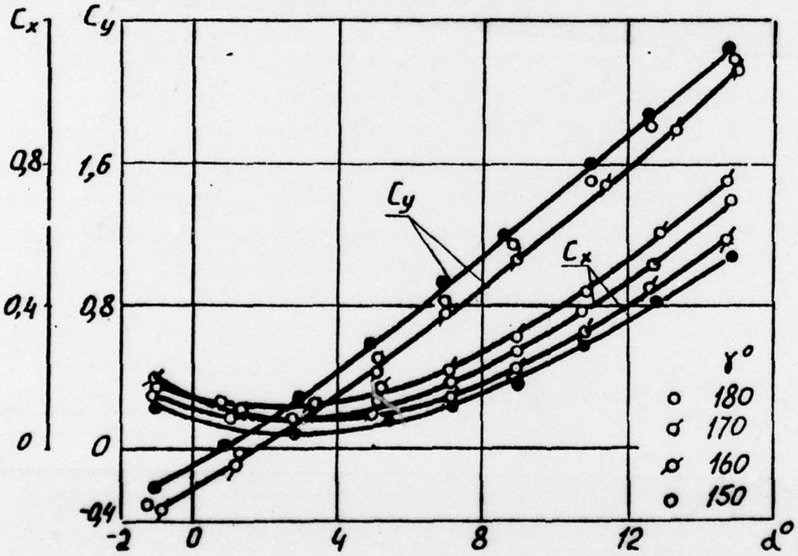


Fig. 10.

Page 68.

The obtained results attest to the fact that with the decrease of the expansion angle of V-shaped wing the value of drag coefficient from the decrease of expansion angle falls. As a result lift-drag ratio with decrease of angle γ from 180° to 150° increases by 20-30o/o in comparison with the maximum quality of delta wing (Fig. 11). Since the amount of lift does not in effect depend on the expansion angle of wing (Fig. 10), also in the class of the wing equivalents with the constant value of lift, the decrease of expansion angle will lead to the increase of lift-drag ratio.

Center of pressure in limits of accuracy of experiment is not virtually changed during a change in the expansion angle of wing and it is arrange/located from the apex/vertex of wing to its 0.68 mean chord, which will coincide with the results of the tests of cones and delta wings at supersonic speeds of flow.

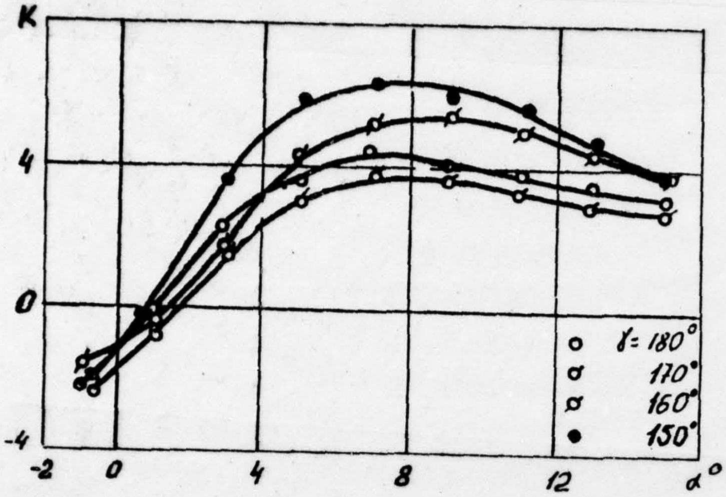


Fig. 11.

Page 69.

In conclusion let us emphasize that an increase in lift-drag ratio of V-shaped wing with the decrease of angle γ occurs not due to the increase of airfoil lift, while in essence as a result of the decrease of its resistance; i.e., appears the same effect, which leads to the low values of resistance of body with the star-shaped form of cross section.

REFERENCES

1. Kuchta B.J., Spencer B.Jr., Aerodynamic and flight characteristics of several variable geometry entry spacecraft during subsonic wing deployment and lauding.AIAA, Paper N 742, 1969.

2. G. I. Maykapar. On the wave impedance of axially nonsymmetric bodies at supersonic speeds, PMM, Vol. 23, iss. 2, 1959.

3. Nonweiler T., Aerodynamic problems of manned space vehicles. J.Roy. Aeron.Soc. 63, 521-528, 1959.

4. Nonweiler T., Delta wings of shapes amenable to exact shock - wave theory, A.R.C., v.22, 644, 1961.

5. Sukes D.M, A note on the supersonic flow past plane and inverted V delta wings, A.R.C., v 22, 848, 1961.

6. Peckham D.H., On three-dimensional Bodies of delta planform which can support plane attached shock waves, A.R.C., C.P. N 640, 1962.

7. Squire L.C., Pressure distributions and flow patterns at M = 4,0 on some delta wings of inverted. V cross sections, R.Q. E., T.N. N Aero 2838, A.R.C., 24449, 1962.

8. Towneed L.H., On lifting bodies of delta planform which can support plane attached shock waves. A.R.C. 25277, 1963.

9. Pennelegion L., Cash R.F., Preliminary measurements in shock tunnel of shock angle undersurface pressure related to a Nonweiler wing. A.R.C., C.P. N 684, 1963.

10. V. V. Keldysh. Exact solutions for carrier systems with one and two step shocks of packing/seal, Eng. Journal. Vol. 1, iss. 3, 1961.

11. G. I. Maykapar. On the construction of supersonic flow near the

flow around solid bodies with the aid of the step shocks of packing/seal. Izv. of the AS USSR, mechanics and machine-building, No 5, 1964.

12. V. V. Keldysh. Study of flow in vicinity of V-shaped wings, formed by stream surfaces behind the step shock of packing/seal. IZV of the AS USSR, MZhG, No 4, 1967.

13. G. Kamimoto and I. Uenaka, Measurements of the hupersonic aerodynamic characteristics for several kinds of wave riders. Dep. of aero. engin., Kyoto univer., C.P. N 25, 1968.

14. V. V. Keldysh, G. I. Maykapar. Gas-dynamic construction of hypersonic aircraft. Izv. of the AS USSR, MZhG, No 3, 1969.

15. A. L. Gonor. The exact solution of the task of the flow around some three-dimensional bodies of the supersonic flow of gas. PMM, Vol. 28, V. 5, 1964.

16. A. L. Gonor. Some three-dimensional/space flows with Mach interaction of shock waves. Report on the 7th symposium in contemporary problems and according to methods in hydromechanics., Jurata, Poland, 1965. (MZhG, No 6, 1966).

17. A. L. Gonor, A. I. Shvets. Investigation of the distribution of pressure on some star-shaped bodies with mach number 4., PMTF, No 6, 1965.

18. A. L. Gonor, A. I. Shvets. Study of the shock envelope during the flow around star-shaped bodies. Izv. of the AS USSR, MZhG, No 3, 1966.

19. A. L. Gonor, A. I. Shvets. Flow around V-shaped wings with number = 4, Izv. of the AS USSR, MZhG, No 6, 1967.

20. Küchemann D., Hypersonic aircraft and their aerodynamic problems, Progress aeron. Sci, Perg. Press, v. 6, 1965.

21. A. L. Gonor. Law of resistance of Newton for the bodies, formed by the intersecting surfaces. Izv. of the AS USSR, MZhG, No 1, 1967.

22. A. L. Gonor. Some three-dimensional/space flows with flywheel interaction of shock waves. Izv. of the AS USSR, MZhG, No 6, 1966.

DISTRIBUTION LIST

DISTRIBUTION DIRECT TO RECIPIENT

ORGANIZATION		MICROFICHE	ORGANIZATION		MICROFICHE
A205	DMATC	1	E053	AF/INAKA	1
A210	DMAAC	2	E017	AF/RDXTR-W	1
B344	DIA/RDS-3C	9	E403	AFSC/INA	1
C043	USAMIIA	1	E404	AEDC	1
C509	BALLISTIC RES LABS	1	E408	AFWL	1
C510	AIR MOBILITY R&D LAB/FIO	1	E410	ADTC	1
C513	PICATINNY ARSENAL	1	E413	ESD	2
C535	AVIATION SYS COMD	1		FTD	
C591	FSTC	5		CCN	1
C619	MIA REDSTONE	1		ASD/FTD/NICD	3
D008	NISC	1		NIA/PHS	1
H300	USAICE (USAREUR)	1		NICD	2
P005	ERDA	1			
P005	CIA/CRS/ADB/SD	1			
NAVORDSTA (50L)		1			
NASA/KSI		1			
AFIT/LD		1			

FTD-ID(RS)T-1017-78